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REPORT ON CERTAIN ASPECTS OF SPACE ENVIRONMENT SIMULATION

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On the Problems of Aircraft and Spacecraft Structure

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Various space environment simulators, including NASA, General Electric, Jet Propulsion Laboratory units, are reviewed, with brief description of their function and expected life. The Plum Brook and Mark projects are evaluated, and the simulator of the French Subsidiary of Thomson-Houston is described. A combination of vibration, temperature, and vacuum tests is suggested, for more accurate failure detection. The importance of accurate estimates of energy and heat dissipation, inside the satellite, thermal gradients, energy loss by radiation of the satellite, and other environmental parameters is stressed and demonstrated by tabulated data. author

INTRODUCTION

The estimated approximate cost price of 100,000 dollars per kg of space vehicle placed into orbit [see J.C.New, Chief, Test and Evaluation Division of Goddard Space Flight Center, GSFC; (Bibl.1)] emphasizes the extreme importance of ensuring maximum success for orbital flights, intimately connected with the quality of all equipment and components as well as with the nature of the tests. A realization of such tests under space environment conditions thus constitutes a basic prerequisite. However, space simulation encounters two fundamental difficulties, one of which is of a technical nature since space conditions can only

* Numbers in the margin indicate pagination in the original foreign text.

be approximated since they must be reproduced within necessarily confined enclosures, while the other is of a financial nature since the cost of actual development is high. This cost should not exceed 20% of the overall cost, so as to prevent jeopardizing the space programs themselves.

Although all simulators of space environment primarily are vacuum enclosures, it is obvious that application of vacuum technology is not restricted to merely designing vacuum chambers with a given pumping system. Rather, a simulation of the phenomena under study and of their performance requires the development of entire vacuum complexes, designed for the utilization of various techniques which frequently are contradictory and are difficult to adapt to anticipatory studies.

For this reason, vacuum technology always has been one of the first beneficiaries of the scientific and industrial progress induced by space research, as was clearly demonstrated during the First International Congress on Vacuum Technology in Space Research (CIVRES), held in Paris in June 1964.

Basing our further discussion on the papers presented at this Congress, /2
we will attempt to formulate the problem of simulation, specifically with respect to satellites, by first reviewing the inaccuracy of definition of some of the parameters and then giving an overall view of the methods and means generally used and the difficulties encountered. In addition, some practical results will be given.

1. Purpose of Space Environment Simulation

The launching of space vehicles is closely dependent on the quality of the entity of materials and structures and on their control, under conditions approaching space environment as closely as possible.

Increasingly high vacuum as a function of altitude (Fig.1) (Bibl.2). Up to 90 km (10^{-3} torr*), the pressure variation can be represented by a straight line.

Modification in the type of atmosphere composition (Fig.2) (Bibl.3, 4).

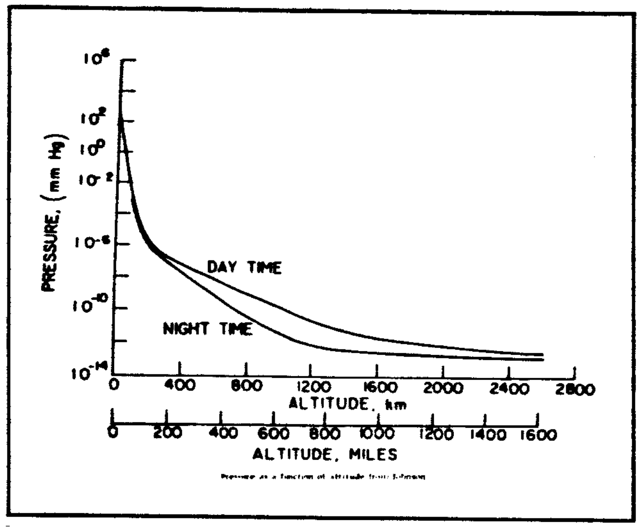


Fig.1 Pressure as a Function of Altitude

Thus, the principal constituents are as follows:

atomic oxygen from about 250 - 700 km, i.e., $\sim 1 \times 10^{-6}$ to 10^{-9} torr;

helium from about 700 - 2000 km, i.e., $\sim 10^{-9}$ to 10^{-12} torr;

atomic hydrogen, above about 2000 km.

Extreme cold of space: The radiation received by a spacecraft corresponds to that of a black body at about 4°K.

Electromagnetic radiation, primarily of the solar type, with the corresponding albedo in the vicinity of the earth and its characteristic infrared radiation. /3

* The torr is the recommended term for mm Hg, equal to 133 newton/m^2, which is the legal unit in France under the name of "pascal".

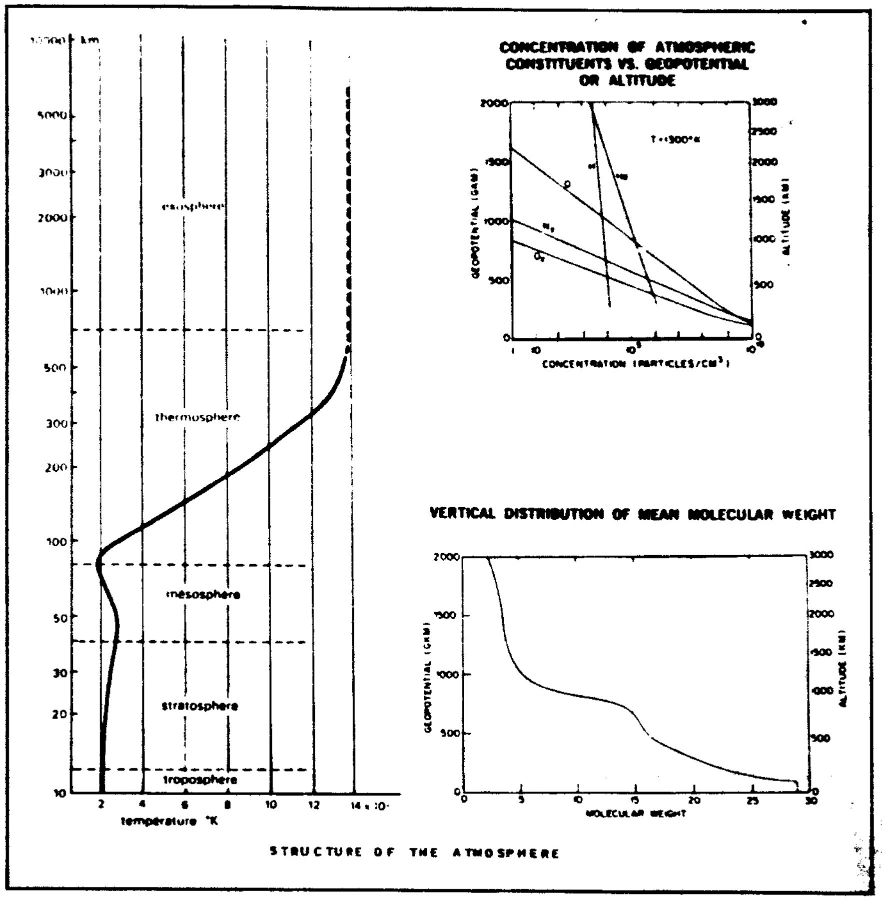


Fig.2 Temperature, Composition, and Molecular Mass of the Air as a Function of Altitude

Exposure to low pressures is one of the main purposes of simulation. Thinking specifically of thermal insulation, it becomes immediately obvious that the thermal problems are closely connected with such insulation and thus constitute the main basis of space simulation. The useful life of a satellite is directly linked with the equilibrium temperature which it will assume in orbit. Thus, the main objective of space simulation is making certain, before launching, that this equilibrium temperature will remain within a range compatible with the functioning of certain critical components, i.e., in general, near 300° K.

Conversely, it is impossible to evaluate a heat balance by experiment unless a sufficiently rarefied atmosphere is available; this must be below 10^{-5} torr at which pressure heat exchange proceeds only by radiation. This makes obvious the important role played by vacuum technology in space simulation.

Thus, according to statistics (Fig.3) published by NASA and referring to five satellites of the Explorer type, 51% of the failures are revealed in vacuum tests (Bibl.5).

SPACECRAFT GROUND TEST EXPERIENCE
(AVERAGE OF 5 EXPLORER TYPE SATELLITES)

	Elec.Failures		Mech.Failures		Total	
Test	No.	%	No.	%	No.	%
Checkout	2.4	13	1.2	26	3.6	16
Vibration	4.0	22	2.8	61	6.8	30
Temperature	.6	3	-	-	.6	3
Vacuum	1.0	6	-	-	1.0	4
Thermal Vacuum	10.2	56	.6	13	10.8	47
Total	18.2	100	4.6	100	22.8	100

NASA SD63-1448

Fig.3 NASA Statistics on Five Satellite Tests

In space, a vehicle acquires an equilibrium temperature resulting from the following balance:

Energy component due to radiation absorbed by the vehicle and to dissipation of the internal energy sources.

Loss of energy radiated by the vehicle at its prevailing temperature.

Solar radiation corresponds to a mean energy E_s of 1.4 kw/m^2, uniformly 14
distributed with a decollimation angle of 32' and whose spectral distribution is close to that of a black body at a temperature of 6000° K.

Radiation due to diffuse reflection of solar radiation by the earth and its atmosphere constitute the terrestrial albedo. On the average, this radiation is equal to 35% of the solar radiation, but recent measurements have shown variations in the albedo from 0.27 to 0.60, depending on the position of the satellite, on passing from the tropics to about 45° latitude.

The infrared radiation due to the earth and its atmosphere known to correspond to a black body of about 250° K, i.e., 224 w/m^2, decreases in inverse proportion to the distance, as the albedo.

Thus, the energy effectively received by the satellite depends on its position, as shown in the following Table, compiled as a function of the angle between the earth - satellite and earth - sun axes (Bibl.6).

TABLE

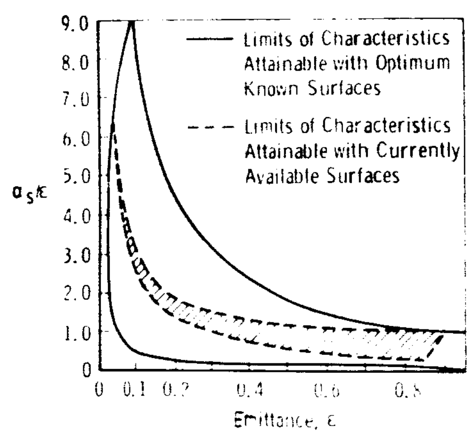
RADIATION ENERGIES AT VARIOUS POSITIONS OF AN EARTH SATELLITE

Angle Made by the Earth - Satellite and Earth - Sun Axes	E_{sun} (w/cm^2)	E_{alb} (w/cm^2)	E_{pla} (w/cm^2)	E_{tot} (w/cm^2)
0°	1396	500	224	2120
30°	1396	410	224	2030
60°	1396	225	224	1845
90°	1396	0	224	1620
180°	0	0	224	224

This introduces a factor which, in addition, depends closely on the geometric configuration of the satellite wall and on its accessories and which has to do with the considered radiation. Similarly, a differentiation must be made between the area of the satellite exposed to solar radiation and that subject to albedo and earthshine.

It is of importance to mention that an accurate evaluation of the energy /5 received by a given satellite is extremely difficult despite the use of programing devices and electronic computers.

Finally, the absorption factor α and the emittance ϵ must also be considered. Although, conventionally, their ratio is taken as equal to unity (perfect black body) since one refers then to equal spectral distribution of both absorbed and emitted radiation, the value of this ratio in the case of spacecraft may vary within a wide range, as shown by the curve in Fig.4 which was plotted



Limits of thermal control with available radiation surfaces

(from Smelt)

Fig.4 Limits of Thermal Control with Available Radiation Surfaces

by Goethert (Bibl.7) for ideal surfaces and for surfaces with coatings that are readily reproducible at the interior of a spacecraft. In fact, in the case of solar radiation, the quantity α_s corresponds to black-body radiation at 6000° K whereas the radiation emitted by the spacecraft corresponds to that of a surface at an emittance ϵ and at a mean temperature of 300° K.

The existence of any attitude stabilization other than that obtained by rotation of the moving body, completely changes the distribution of the thermal gradients.

The next point to be considered is the internal energy dissipated within the satellite itself since, even if its value is accurately known in principle, an error of any sort might be far-reaching. Porter (Bibl.8) mentioned the case of a satellite in which the battery froze because of a wrong estimate of the interior energy dissipation of the satellite, giving a value of 10 w instead of 4 w, which could have been avoided if the tests would have been more comprehensive.

The complexity of the parameters, having to do with thermal problems, does not exactly facilitate the attempts at space simulation because of the inaccuracy resulting from the difficulty in their theoretical determination. Thus, the prime goal must be an evaluation of the relative significance of the space /6
environment parameters and of their value. However, this is a task which can be attacked and solved only on the basis of experiments.

2. Methods of Space Environment Simulation

The complexity of an overall theoretical evaluation of the heat balance necessitates a large number of tests, from the stage of layout of the materiel to an analysis of the final behavior of a spacecraft.

An ideal simulator must satisfy the following conditions:

Expose the satellite to radiations close to those experienced in space.

Simulate the absorption of all radiations emerging from the satellite, i.e., attempt to simulate the relatively infinite heat trap of space.

Reproduce the sorption of all molecules emerging from the satellite, i.e., attempt to reproduce the practically infinite pumping characteristics of space.

The simulator, kept under vacuum, thus will be characterized by a cryostatic wall or cryoplates which could be called a "space wall" and which would thus play a dual role:

1) The first function is that of contributing to pumping of the simulator chamber by furnishing, beyond a certain pressure, the largest portion of the pumping capacity, specifically relative to gases that condense at the temperature of this wall, thus ensuring homogeneous pumping conditions with respect to the vehicle.

In the largest simulator of the General Electric Co., the capacity of /7
the diffusion pumps is about 25,000 ltr/sec, while that of the cryogenic pumping is about 30 million ltr/sec.

2) The second function of the space wall is to absorb thermal radiations, those emitted or reflected by the satellite and those indirectly produced by the use of radiation simulators which, in themselves, are not radiation sources.

In this respect, the temperature of space of 4°K need not be reproduced since the corresponding radiation, already at 78°K, constitutes only 0.5% of the satellite radiation at a mean temperature of 300°K, in view of the fact that the temperature, because of Stefan's law, enters in the fourth power.

Thus, the absorption coefficient α_c of the cryoplate must be as close to unity as possible. To obtain this, blackened structures are used.

However, the required value of α_c cannot be considered independently of the respective dimensions of simulator and satellite. The efficiency of heat traps can be primarily estimated from the error in α_c, determined as a function of α_c and of the ratio of simulator diameter D_c to satellite diameter D_s, as shown in Fig.5 taken from Destivelle's report (Bibl.9).

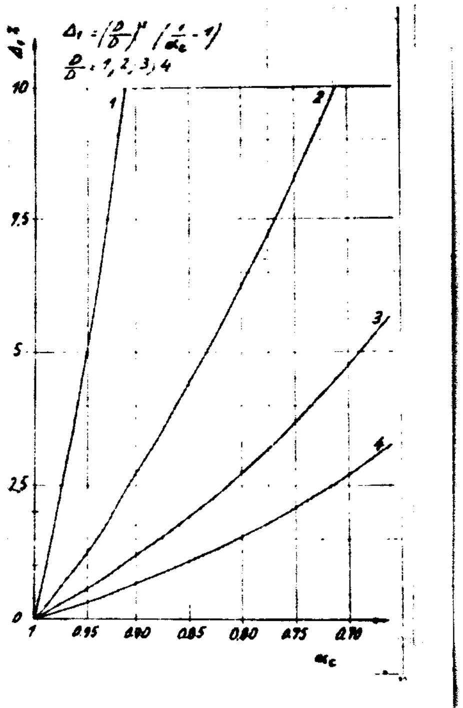


Fig.5 Error of the Absorption Coefficient α_c of the Cryoplates, as a Function of α_c and of the Ratio of Simulator Diameter to Space-Vehicle Diameter

This indicates the importance, from the cost angle, to give values of > 0.9 to α_c so as to reduce the dimensions of the simulator for a given satellite.

3. Radiation Simulators /8

a) Solar Spectrum

It is quite impossible to discuss here the respective merits of artificial power sources, which have been described frequently in the literature. On the spectral plane and without correction, the conventional carbon arc has the optimum characteristics. Xenon arc lamps have an excellent distribution for the spectrum region representing 60% of the total energy (< 0.8 μ). However, the presence of intense lines, centered near 1 μ, may require the use of filters.

Considering the use conditions of such sources, xenon lamps which have probable lifetimes of several hundreds of hours and do not require special precautions in service, seem the most practical. Conventional carbon arcs, even with automatic recharge of the electrodes, apparently do not exceed a continuous operating range of more than 24 hrs, with the additional handicap of a spectral evolution due to the progressive fouling of the optical system.

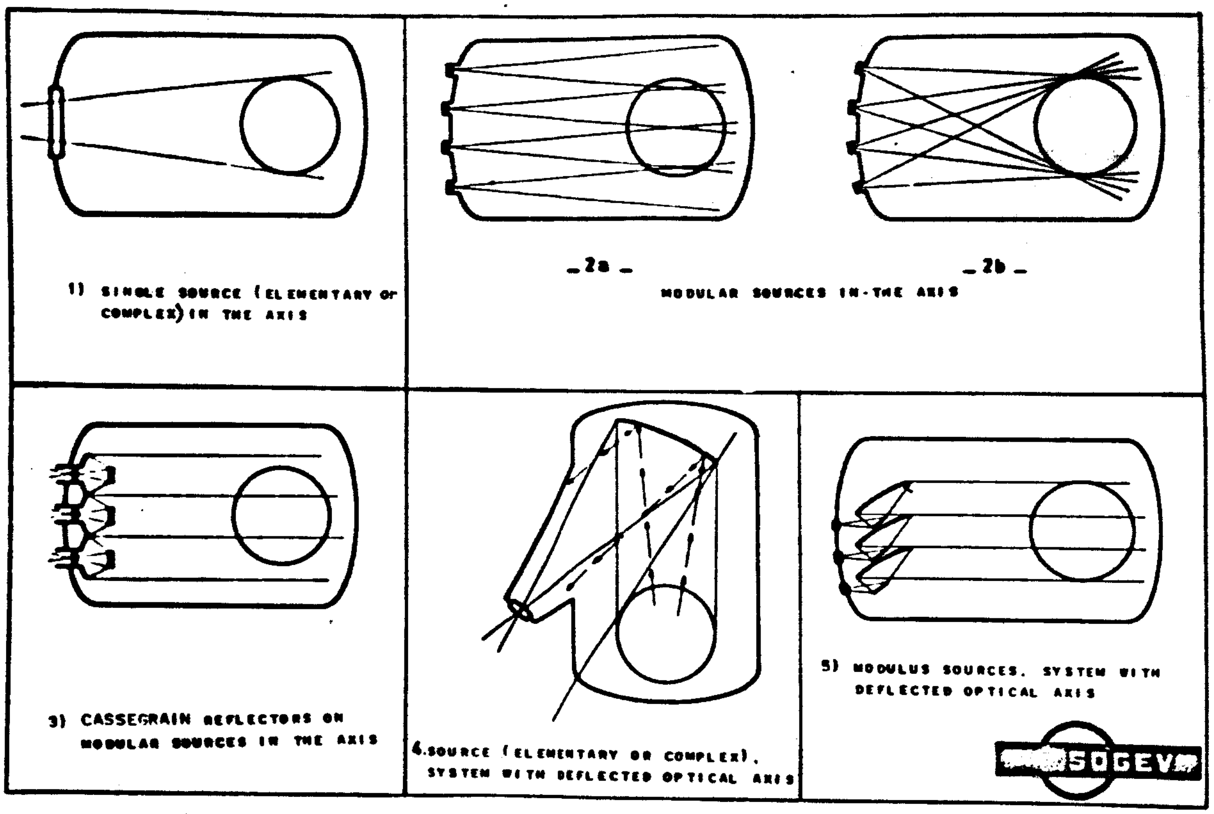


Fig.6 Optical Diagrams for Solar Simulation

The other principal parameters, characteristic of the sources, are decollimation, intensity, and uniformity in plan and in depth. In addition, the perturbations produced by the optical system in establishing the heat balance must be allowed for.

Usually, the sources are placed outside the simulator; to satisfy the intensity required for illuminating the most important surfaces, complex systems are needed which consume power and frequently may limit the overall yield to a value of 5% of the power supplied to the light sources. This requires a large number of lamps.

The optical systems are schematically shown in Fig.6.

At the large simulated solar beams used, decollimations of a few degrees are already difficult to obtain. Lee and Steg (Bibl.10) gave an estimate of the temperature error for the heat balance, as a function of the decollimation angle (Fig.7). /9

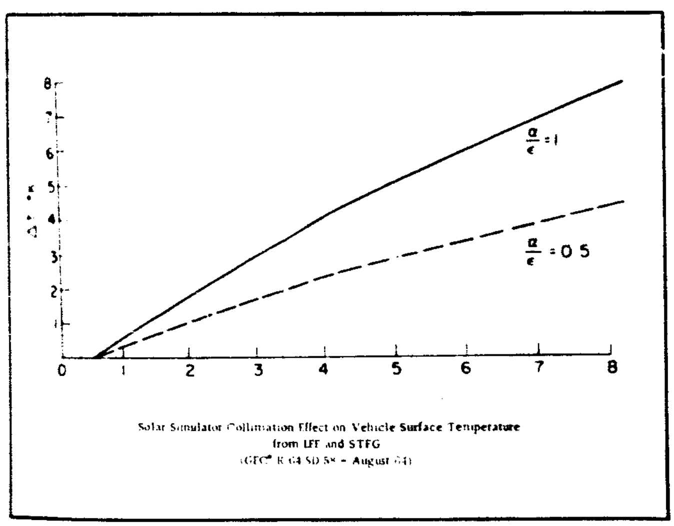


Fig.7 Temperature Error on the Surface of a Model, as a Function of the Decollimation Angle

The maximum possible uniformity, in the optimum case, may reach about ±5% in the setup under consideration.

The optical devices within the chamber, seen by the satellite, must be cooled to sufficiently low temperatures, depending on their characteristic emissivity, so as to prevent them from becoming infrared emission sources themselves. This is specifically the case for mirrors of systems with shortened optical axis.

If spectral fidelity is not required or if it can be readily compensated, the thermal flux can be simulated by incandescent lamps, infrared lamps, heating filaments, or - more generally - by a plate maintained at the required temperature.

4. Simulation of Albedo and Earthshine

In view of the variations in the characteristics of these radiations, this type of simulation is extremely delicate and involves bulky equipment. Therefore, one is frequently satisfied with establishing a mean radiation value from which, by analytical considerations, the heat balance is extrapolated for other values, applying estimated corrections to the committed errors.

This type of simulation is closely connected with the reproduction of the orbital satellite motion. If the satellite is stationary, the infrared source must be shifted either by mechanical means or by programing the operation of a certain number of heating elements within a larger overall unit.

If the satellite is moving with two degrees of freedom, this simulation /10
can be made integral with one of the motions, as presented in a paper at CIVRES by Latvala and Peters (Bibl.11) and shown in Fig.8.

Simulation of the spectral distribution of the albedo is even more difficult and requires the additional use of xenon lamps.

It is obvious that such simulations are directly linked to the motion and to the type of scheduled attitude stabilization of the satellite and to its design, specifically to its thermal time constant.

a) <u>Vacuum</u>

The problem encountered in producing low pressures within enclosures whose dimensions are larger than ever before, such as the sphere of 30 m diameter

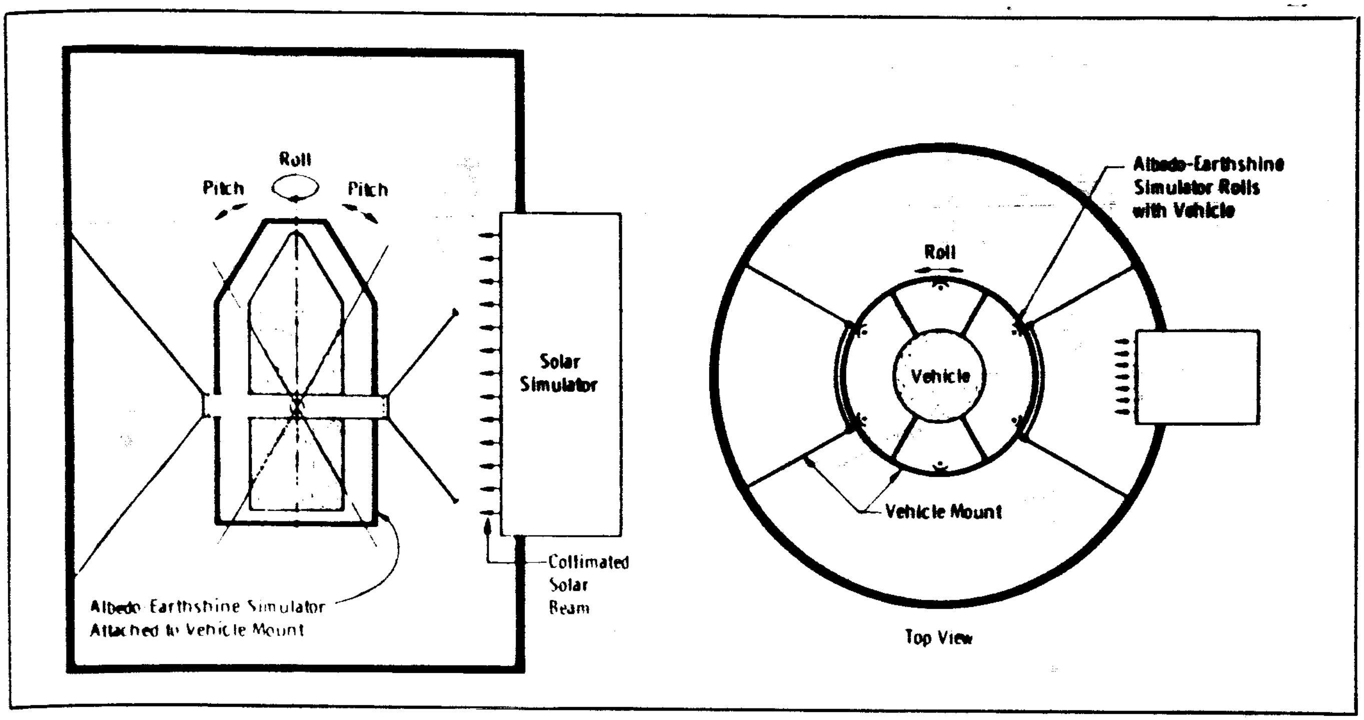
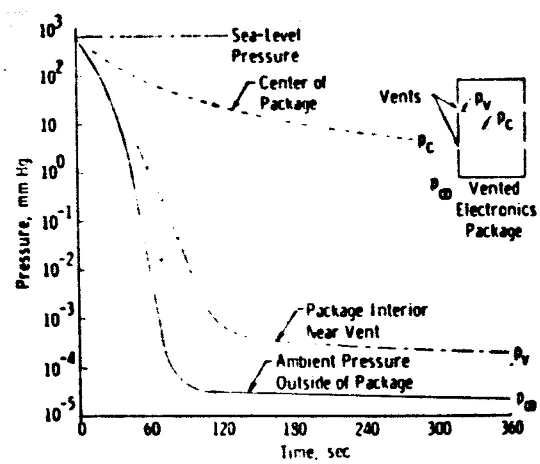


Fig.8 Albedo Simulator (see Latvala and Peters)

used at Langley Field (NASA) where a vacuum of 10^{-6} torr is obtained within 10 hrs (Bibl.12), shows the significance of vacuum technology in space problems.

In the case of complete simulators, the production of low pressures is thus closely connected with the search for solutions that satisfy the requirements of cryogenic and optical technology. At the other end of the scale, the launching of satellites involves numerous mechanical stresses and raises the problem of data transmission from the satellite.

However, the satellite itself may have a time constant with respect to its pumping. In general, satellites are not vacuumtight and the time to obtain a given pressure at the interior of the satellite differs considerably from that expected from the pressure outside the satellite.



Launch trajectory pressure data from simulation test and free flight

(from Cook and Goethert)

Fig.9 Launch Trajectory Pressure from Simulation Tests and Free Flight

Thus, for the testing of electric equipment for resistance to breakdown /11 and electric discharge, it is superfluous to require a pressure of 10^{-5} torr. It is true that this pressure is necessary, but only inside the electronic subunits; in addition, the time required for obtaining this partial vacuum differs completely from the time for obtaining the pressure characteristics (Bibl.7) of the simulator (Fig.9).

Incidentally, the pressure determination is one of the difficulties encountered in the simulation of space vacuum within the simulators.

In fact, the vacuum within a closed chamber cannot reproduce true space conditions, for the following reasons:

> Primarily, no molecule leaving the satellite will ever have a chance to return to the latter whereas, in a vacuum enclosure no matter what the mean free path or the efficiency of the cryoplate might be, a certain number of molecules will return to the satellite.
>
> Secondly, the nature of the residual gases, which might be dissociated and ionized, is an interfering factor.

The presence of a directive molecular flux may lead to a noticeable variation in pressure inside the simulator, for example, at a ratio of 100. Thus, the measured pressures, depending on the conditions of pressure production, may differ greatly from the pressures actually prevailing in the environment of the satellite under test.

Here, we can only briefly review the conventional means for obtaining low pressures. The characteristics depend on the use conditions of the simulators and on the duration of the individual cycles: Even if the tests are of long duration, the advantage of rapidly establishing the forevacuum is obvious. For a full-scale prototype, the statistical mean reveals about 30 problems encountered during the tests [see New (Bibl.13)].

Roots vane-type pumps or also steam-jet ejectors are frequently used for supplementing the primary mechanical pumps. One specific case should be mentioned here, namely, the 60 m^3 chamber (Fig.10) of the National Space Research Administration (France), where a pressure of 5×10^{-6} torr is obtained within /12 about 45 min, after blackening the cryoplate maintained at ambient temperature. The secondary pumping, as is conventional, is done by diffusion pumps equipped

Fig.10 Space Simulator of the CNES (France)

with baffles and liquid-air traps, which are indispensable for cryogenic pumping.

5. Space Simulators

This term includes not only simulators meant for satellite tests but frequently also any chamber used for any test connected with studying equipment or reliability of the components. This is the reason for the difficulty in establishing any classification.

Gaumer (Bibl.14) gave a classification of small chambers, medium chambers, and a third category reserved for tests of full-scale satellites which, for this reason, does not exclusively comprise large chambers.

Unfortunately, it is possible to discuss only US designs because of the complete absence of Soviet publications in this field and because of the present lag in the European potential with respect to large chambers.

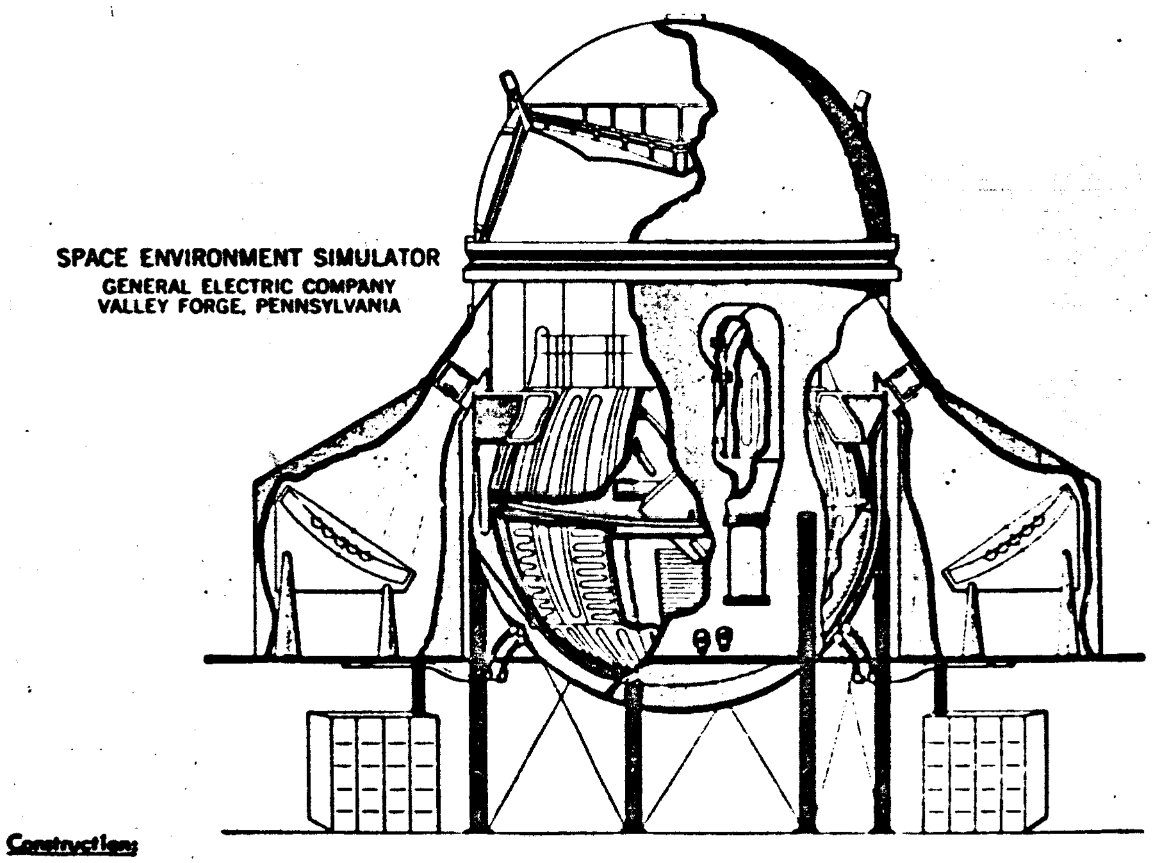


Fig.11 Large Simulator of the General Electric

According to the review presented by Gaumer at the CIVRES, the USA has at least 40 simulators of large dimensions, which generally serve for satellite tests and thus are primarily used for thermal experiments.

The simulators have either a spherical shape or constitute vertical or horizontal cylinders, frequently equipped with facilities for cryogenic pumping, occasionally using helium gas. The most popular pumps are diffusion pumps of 80 cm diameter, having capacities as high as several hundred thousand of liters per second. The extreme pressures reported for vacuum chambers go as high as 10^{-9} torr. /13

About half of the chambers are equipped with mechanical devices for rotating the satellite.

The solar simulation, produced in one third of the cases, is effected in about half of the chambers by means of high-intensity carbon arcs. This rarely matches the performance of a system with shortened optical axis such as had been obtained in the large General Electric simulator (9.75 m diameter; 16.5 m height) which, apparently, is the most perfected device to date and which has been briefly described above (Fig.11).

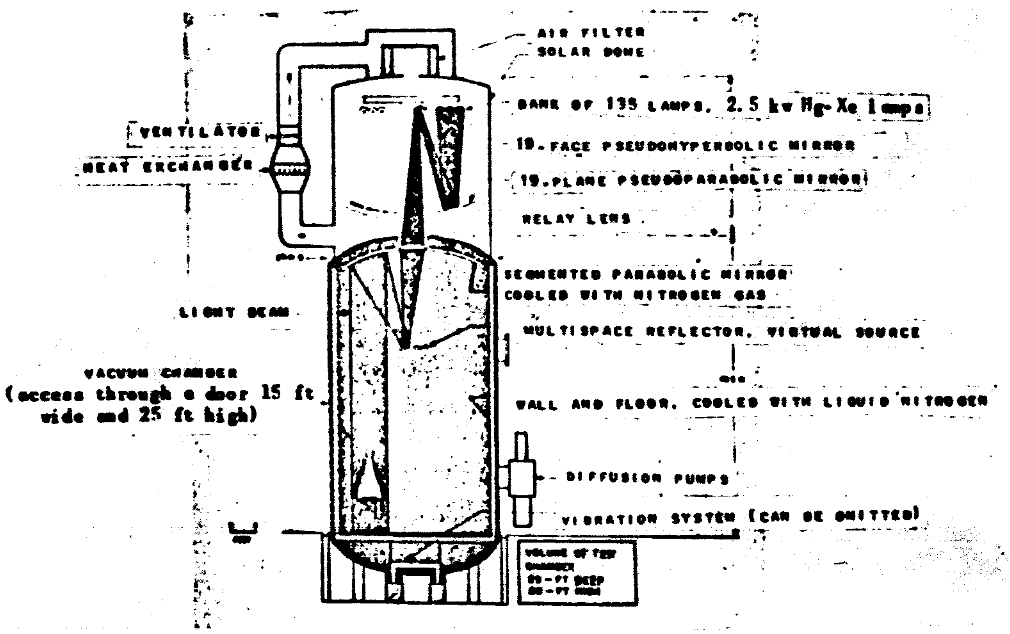


Fig.12 Large Simulator of the Jet Propulsion Laboratory

The testing space has a diameter of 8 m and a height of 6.4 m; the extreme pressure is reported as 10^{-9} torr, using cryogenic pumping with liquid nitrogen and gaseous helium. The characteristics of the solar simulator are as follows: 148 xenon lamps of 5 kw; beam of 5 m diameter; uniformity of 5% per 70% of surface; decollimation angle of 3°. The parabolic reflectors required a long focus, which was finally achieved by using 1560 individually adjustable mosaics

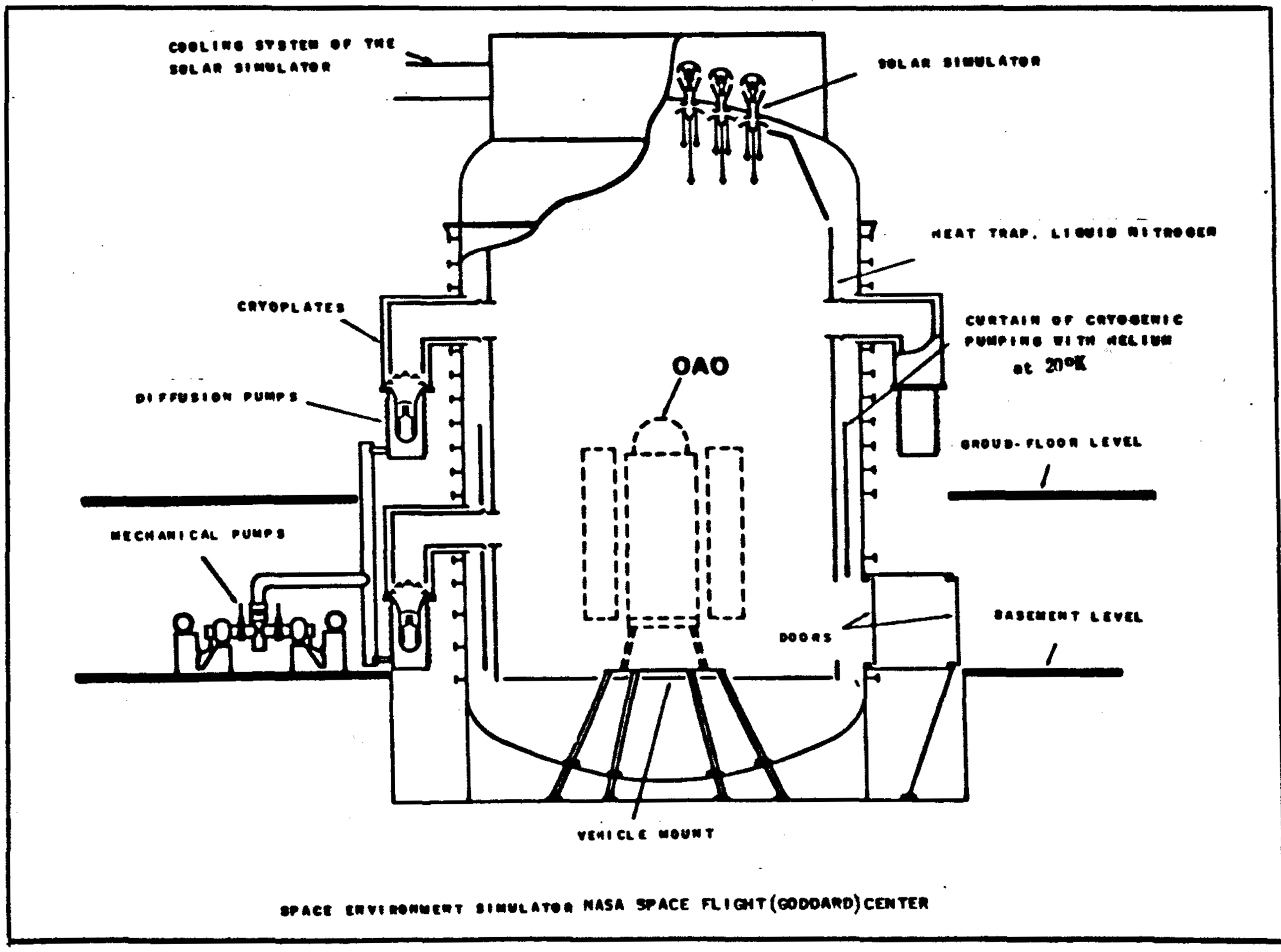


Fig.13 Large Simulator of the Goddard Space Flight Center

whose parameter determination could be obtained only by means of electronic computers (Bibl.15).

The large simulator of the Jet Propulsion Laboratory (Fig.12), having a minimum pressure of 10^{-7} torr with cryogenic pumping with liquid nitrogen, required the development of a new optical system (with a compound source formed by Hg-Xe and Xe lamps) on a smaller simulator (Bibl.16).

The large simulator (Fig.13) of the Goddard Space Flight Center (GSFC) has a limit pressure of 10^{-9} torr with cryogenic pumping with liquid nitrogen and gaseous helium; the testing space is 8 m in diameter and 11.5 m in height while the solar simulation is obtained with a modular source unit, using xenon lamps and furnishing a decollimation of 3.2 to 4.4° (Bibl.17). /14

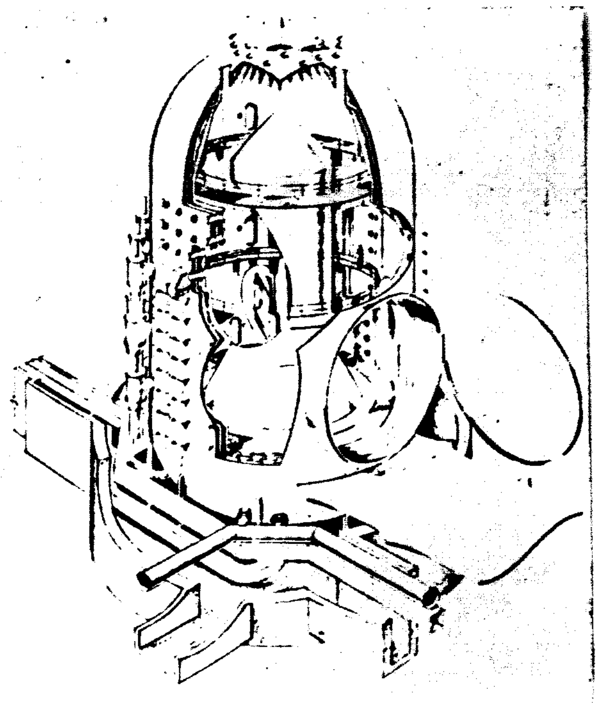


Fig.14 Simulator A of the Manned Space Center (Houston)

It is of interest to mention that, for each of the above simulators with lowest pressures of about 10^{-9} torr, there exist other large simulators intended for partial or rough tests, such as three spheres of 11.5 m diameter (10^{-9} torr) at the General Electric and exact duplicates at the Jet Propulsion Laboratory

(10^{-6} torr) and at the Goddard Center (10^{-3} torr for dynamic tests).

Still other chambers have even greater dimensions, such as the A-chamber of the Houston Manned Spacecraft Center (MSC), used for tests of the Apollo program and having a diameter of 19.5 m and a height of 24 m (Fig.14). This particular chamber is equipped with two solar simulators, having modular carbon-arc sources. The vehicle is placed directly on a revolving platform, simulating the lunar soil and having a temperature that can be varied from 100 - 400° K. This chamber is intended for tests at the human scale and thus had to be equipped with air locks and devices specifically designed for rapid repressurizing in the event of danger to the human crew.

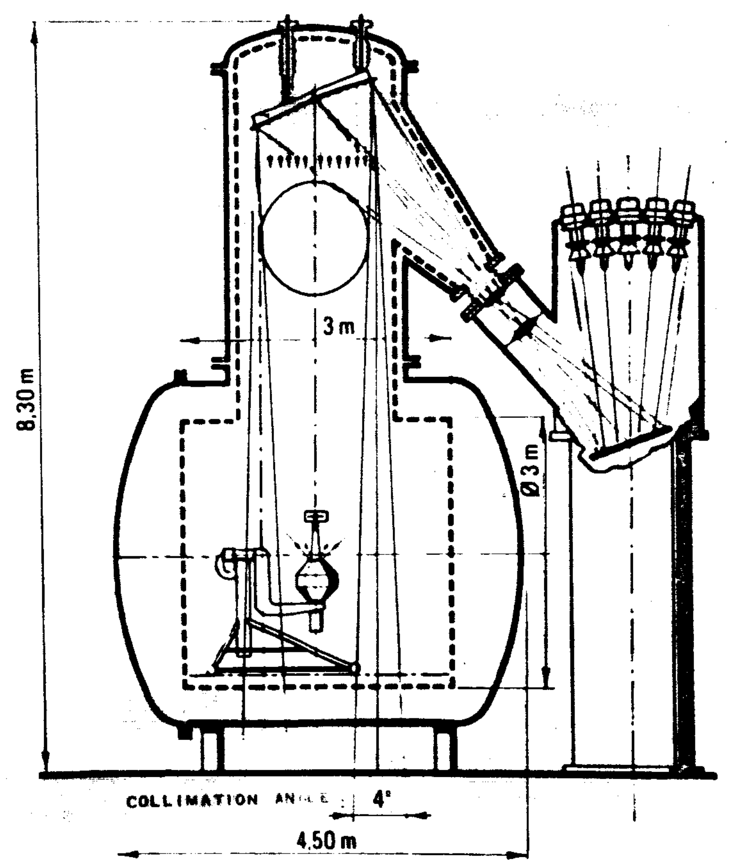


Fig.15 Schematic View of a Space Environment Simulation Chamber for Tests on Small Satellites

Another large chamber is that at the Arnold Engineering Development Center (AEDC), Mark I, having a diameter of 12 m and a height of 22 m and comprising 48 diffusion pumps; this chamber can be connected to a wind tunnel.

Solar simulation is obtained by carbon arcs. The surface of the cryo- /15 plates, cooled by liquid nitrogen and gaseous helium, is about 1300 to 800 m^2, respectively.

This chamber permits an ascent simulation up to an altitude of 24 km (90 sec) and could be used for vibration tests.

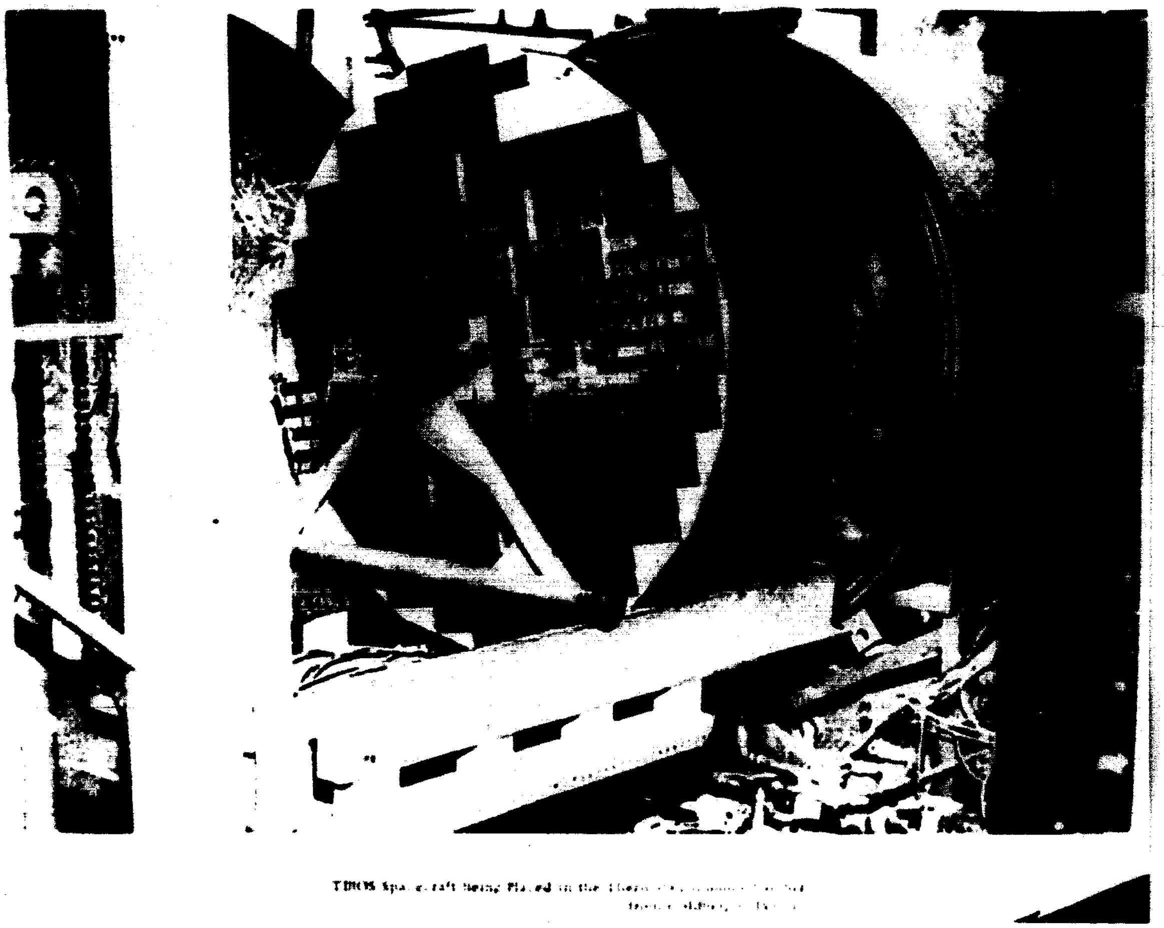


Fig.16 Overall View of the Tiros Satellite in its Simulator

The cost of such chambers may reach many millions of dollars: The cost of the General Electric chamber was six million dollars, that of the Jet Propulsion

/33

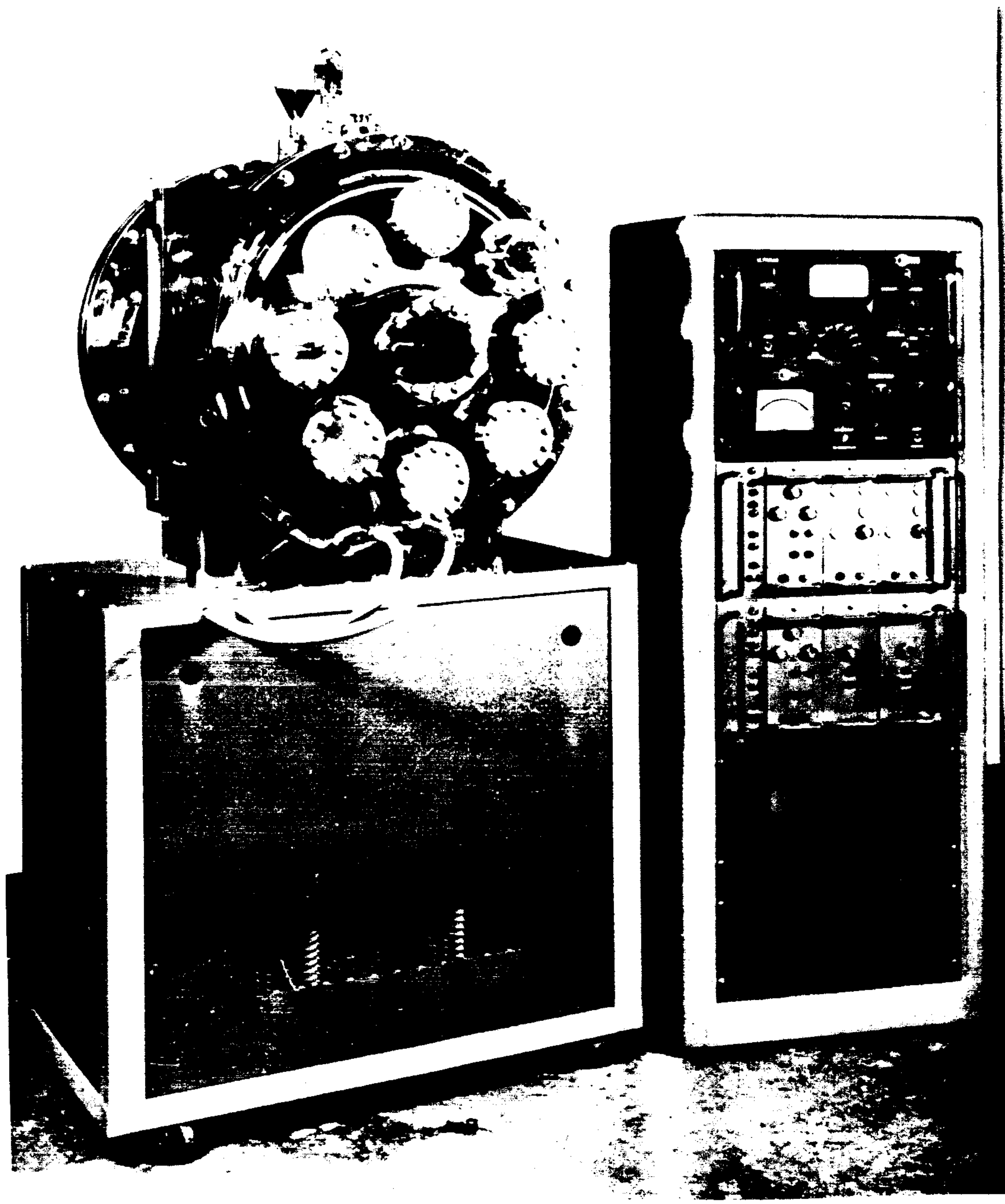


Fig.17 Equipment Constructed by the SOGEV for Component Tests, Made Entirely of Stainless Steel and Equipped with a Cryostatic Screen and a Heating System by Infrared Lamps

Laboratory four million. However, since publication of these figures several years ago, important developments have been added relative to the optical systems.

It can be thus stated that highly "sophisticated" chambers require a long period of development and that the actual yield of their use for satellite tests is not accurately known. Conversely, because of the experimental studies that such chambers permit, they constitute a substantial factor for understanding the fundamental principles of simulation.

Figure 15 gives a schematic view of a somewhat smaller installation, but still yielding a satisfactory ratio of chamber dimensions to satellite dimensions.

Such is no longer the case for still smaller chambers having diameters of only 1 - 2 m; nevertheless, these are useful for thermal tests on full-scale satellites as in the case of Tiros (Fig.16) but not so much for solar simulation (Bibl.18); they are highly convenient for tests on subassemblies and components.

In these latter cases, it is generally not so much a question of reproducing space environment except possibly for studying the efficiency of solar /16
cells which requires as perfect as possible a solar simulation or else for measuring the thermal coefficients of materials or components. It is of advantage to use chambers that permit multiple applications. Figure 17 shows a type of such a chamber, designed by the SOGEV (French Vacuum Corporation).

6. Analysis of Space Simulation and Results Obtained

Since space simulation is at the beginning of its development, it is natural that its fields of application are not yet defined and that even different interpretations are encountered as to conduction of the tests.

For example, one trend is to schedule only definite tests: For example, a test for flight aptitude, if it is conclusive, will permit proper adaptation of the model to its desired end use.

Another trend, conversely, is to make systematic tests on all components, subassemblies, prototypes, and flight models. Such tests will become more frequent with the progressive improvement in quality and reliability of the individual components. So-called check tests on the quality of structure are separate from flight qualification or airworthiness tests.

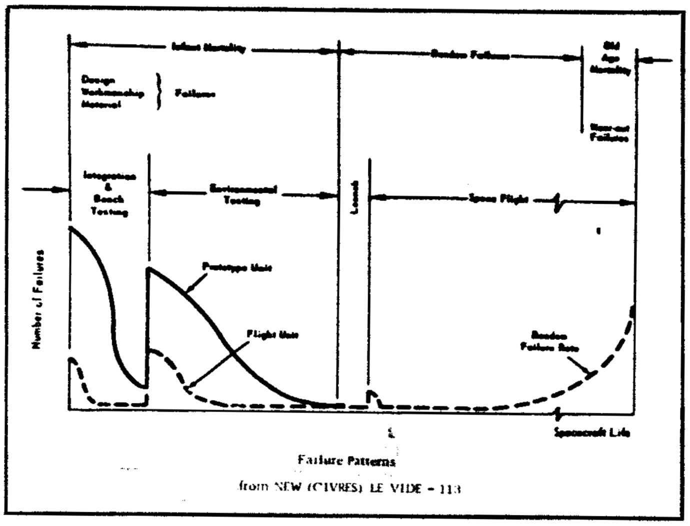


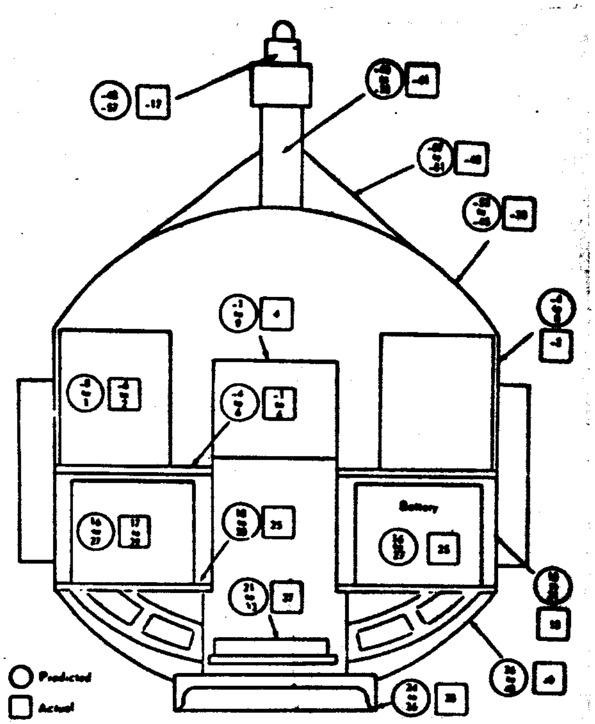
Fig.18 Satellite Failure Pattern

Still another trend is that of testing the overall systems at stress levels determined by changes in the environment over a time sufficiently long to have most failures appear in a detectable form.

For a flight model, the levels must not be reached more than once in /17 20 times whereas, for a prototype, they can be higher; specifically, the test temperature margins can be exceeded by 10° C.

According to statistics by the GSFC, the number of failures detected on prototypes is five times higher than that found in flight models (Bibl.13).

As shown in Fig.18, one main purpose is to detect incipient failure, which is realized in orbital tests lasting about 10 days; conversely, a complete testing program in space environment may last from one month (Explorer XV) to several years (Explorer XVII). For flight models, the duration of the tests (which might be continuous) most often is two months and, on the average, leads to detection of about six problems.



Temperature Predictions Versus Actual Test °C
from Bernier et Al.
(Inst. of Env. Sc. 64)

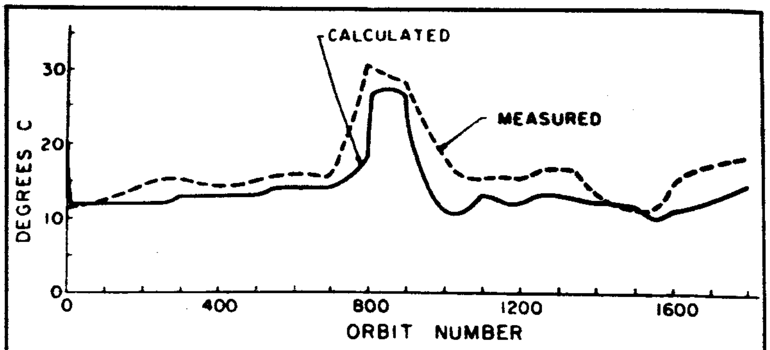
Fig.19 Comparison of Predicted and Measured Temperatures on a Thermal Mockup in a Simulator

In practice, using theoretical and experimental data on the materiel and components and considering a certain number of thermal modules in a satellite model, an attempt is made to obtain a first approximation for first predicting the temperatures within the simulator and then those of the satellite in orbit, as a function of the calculated temperatures.

Thus, the tests in simulators are concerned with a comparison of measured and calculated temperatures, as illustrated in Fig.19.

These tests are basic since they yield detailed data on the equipment to be used in all future tests.

Preliminary tests can be made with a thermal mockup or with reduced-scale models. However, this immediately raises the necessity of an experimental check of the measured values. For example, Fowle et al. (Bibl.20) mentioned a /18



Comparison of Calculated and Measured Component
Temperature on TIROS II Satellite
from Goldberg (CIVRES)

Fig.20 Comparison of Calculated and Measured Component Temperature in Flight of Tiros II

coincidence within 3% for a half-scale mockup and within 10% for a 1/5 mockup.

Other types of thermal tests must also be considered:

<u>Thermal immersion under vacuum</u> within enclosures whose walls are brought to definite maximum and minimum temperatures. From this, the thermal stability of

the vehicle and specifically its time constant is determined which, for example, may be of the order of 20 hrs (Delta rocket). The temperatures of the enclosure are based on the extreme orbital temperatures, so that the temperatures measured on the spacecraft will then correspond to mean maximum or minimum values.

The thermal gradient test comprises the preferential heating or cooling of isolated elements of the total assembly.

The heat balance can best be determined within the framework of solar simulation tests, in which the dynamic temperatures of the spacecraft are measured under assumed operating conditions.

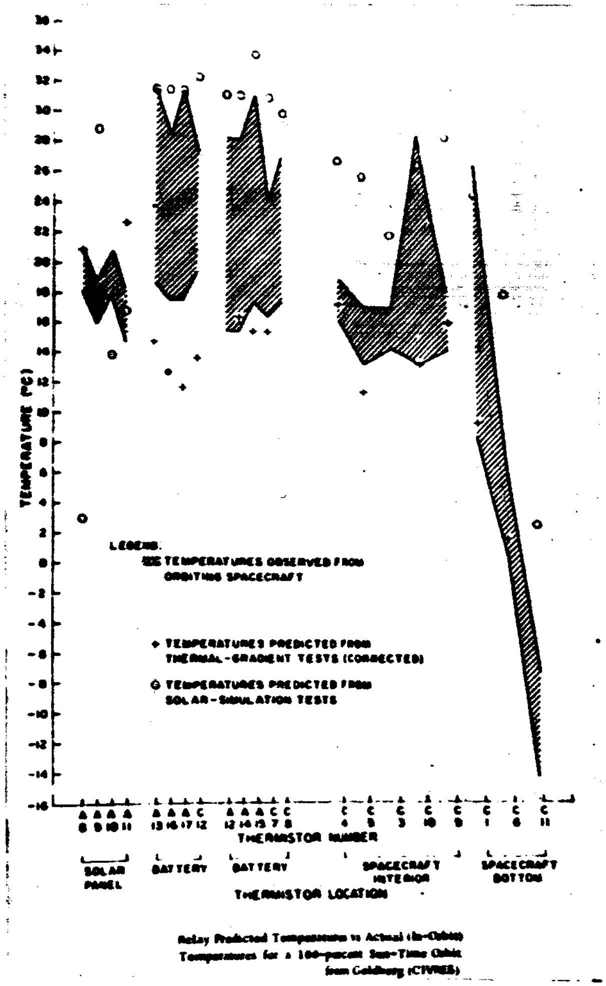


Fig.21 Comparison of Predicted and Measured Temperature in Flight of Relay I

If the satellites are of simple shape, have uniform coatings, and are spin-stabilized, it is possible to determine α/ϵ within the chamber and to adapt the infrared intensity, at equivalent energy, to that of the solar radiation, of the albedo, and of the earthshine, especially if one is able to obtain a certain calibration relative to the conditions prevailing in space, after launching the first satellite of one and the same series. It is of interest to mention the results obtained, for example, on Tiros II by a simple immersion test at extreme wall temperatures of 50°C over a period of three days and of 0°C over a period of five days (Bibl.18), as shown in Fig.20, despite the unfavorable test conditions produced by the ratio of chamber diameter to satellite diameter (see Fig.16), and on Relay II. /19

Space Simulator for Scientific Satellites

Comparison of Temperature Extremes

Spacecraft	Test Temperature, °C		Orbit Temperature, °C	
	Maximum	Minimum	Maximum	Minimum
S-3 (Battery # 2)	41	−5	18	11
S-3a (Battery A)	35	15	21	19
S-3b (Transmitter)	47	2	60	32
S-6 (Center Skin)	40	10	38	8
S-51 Battery A	34	3	47	23
Experiment	34	10	49	7
S-52 Battery A	47	12	49	5
S-74 Transmitter	62	21	54	33
Battery	55	1	52	12
Experiment	38	−1	34	15

Fig.22 Comparison of Critical, Predicted, and Actual Temperatures for Various Satellites of the Goddard Center GSFC

Figure 21 refers to Relay I, tested by thermal gradient experiments and by solar simulation, where the source consisted exclusively of two carbon arcs. In this respect, it will be remembered that Relay I, launched in December 1962, had made about 6000 orbital revolutions by the end of 1964.

For satellites of complex form and heterogeneous surface, excellent results have been obtained (Bibl.13) at the GSFC (Fig.22).

It is not surprising to see that most reports include data on the temperature of the batteries which, actually, constitute the most sensitive elements.

Nevertheless, it should be recalled, as mentioned by Porter, that in about 80 spacecraft successfully placed into orbit in 1963, forty failures occurred in less than two weeks' time; five of these took place during the 14 - 30 first days and two were produced during the 30 following days. Consequently, 50% of these failures were "childhood diseases" which should have been detected on the ground. It will also be recalled that there was a difference of 57°F, observed on Mariner II, when close to Venus.

Considering that "retroactive" analyses usually show up the faults of a given simulator thus making it possible to compensate these, it is obvious that as perfect a simulation as possible would constitute an additional guarantee for success. However, considerable financial difficulties are encountered here. Sufficient data would have to be accumulated to permit an evaluation (among others) of the error produced in determining the equilibrium temperatures of a given satellite during tests in the simulator, on the basis of various faulty performances; this should make it possible to define the reduction in total /20 error, as a function of the cost price.

The extent of the effort made in performing such tests can be appreciated when considering the actual useful life of a large simulator which, for example, was estimated by S.L.Entres as being 70 days in some cases, for a dropout period of 70 weeks (Bibl.21) needed for preparing the chamber, work-up and reduction of the data, changes to be made in the satellite, etc.

CONCLUSIONS

It would be presumptuous and rash to draw conclusions on the topic of a technology that is still in full expansion and, as such, is subject to numerous controversies as revealed in the closing session of the CIVRES (Bibl.22), including: relative usefulness of small or large chambers; necessity or no necessity of the most authentic possible solar simulation; respective estimate of the true cost of analytical methods and of simulation tests which such analyses would make unnecessary; etc.

Therefore, we will restrict our discussion to certain trends, mentioning first that the large simulators, like the small chambers, are being subjected to similar investigations as to their optimum performance and that the progressive improvement in component quality, facilitation of check tests, and accumulation of practical experience in the field of simulation would finally permit a separation of reliability tests from flight aptitude tests.

In view of the high cost of their use, large simulators should thus be reserved primarily for airworthiness tests. It should be emphasized that, although the cost price increases with the size of the simulators, the "size" factor is a useful parameter in obtaining the required accuracy of temperature for the heat balance and for evaluation of the cost. This factor might even /21 become indispensable, from the technical as well as from the economical viewpoint, whenever it is a question of reducing the cryogenic power required for absorption of one kilowatt of radiation reflected by the satellite.

The various reliability tests and the basic research will be performed primarily in special chambers of more reduced dimensions, which leads to the necessity of having a large number of such chambers available.

In fact, there is a tendency to have analytical models replace certain tests, specifically in cases in which their basic data will then be experimentally checked. However, the complexity of shape of satellites and the constant search for better performance naturally constitutes a limit to systematic extrapolations.

1. Combination Tests

The conduction of separate and successive tests in various chambers presupposes that the overall and simultaneous test, relative to the parameters in question, must be equivalent to the individual tests. In reality, such equivalence is far from being realized in all cases. For this reason, it seems that a combination of vibration tests with temperature and vacuum tests would reveal failure patterns that cannot be detected by any of these tests when performed separately (Bibl.23).

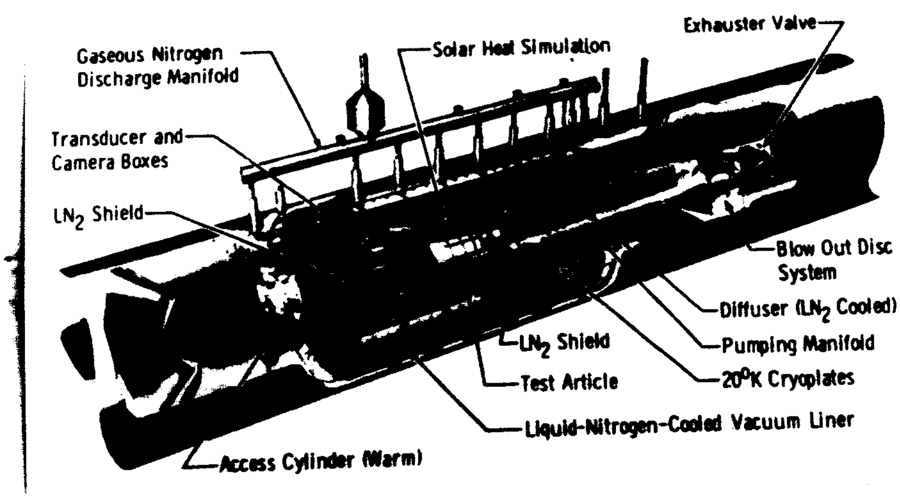


Fig.23 J-2A Cell of the Arnold Engineering Development Center (AEDC)

Space vehicles in orbit require the proper setting of small rockets which must operate at maximum reliability under space environment conditions, with predetermined characteristics and without parasite effect produced by the ejection of jets in the immediate vicinity of the craft.

Equipment developed for this particular effect has been installed at the Arnold Engineering Development Center (AEDC), known as the J-2A cell (Fig.23), provided not only with pumping units comprising mechanical pumps, diffusion pumps, and cryogenic pumps with liquid nitrogen and gaseous helium, but also with cryoplates and infrared lamps for simulating the thermal radiations of space (Bibl.24). /22

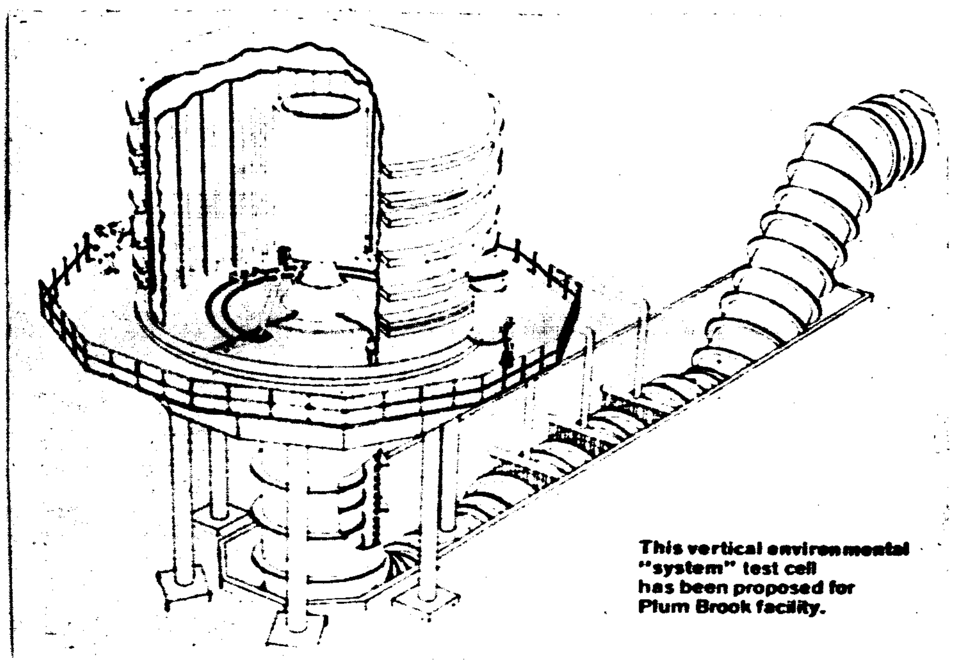


Fig.24 Project of Simulator (Plum Brook)

A combination of chemical propulsion tests with space environment tests would be desirable for improving the design of the upper stage of launch vehicles and is particularly necessary for future lunar missions. This is the main object of the chamber project of Plum Brook (Bibl.24), shown in Fig.24.

For space simulation, required for self-correcting attitude systems, a project was developed in which, starting from a certain pressure, the pumping capacity of the large Mark I simulator is used for simulating the ascending phase of orbit insertion of a satellite and for permitting the proper firing of small rockets.

Such a project emphasizes the interest in auxiliary resources offered by the large simulators. Within the framework of complex units, such simulators also can be used for supplementary aerodynamic tests. The inherent difficulty of using such large simulators lies in the fact that time for building such simulators is equal or even longer than that required for building the satellites themselves. A general trend in the USA consists in escalating the construction of very large chambers, or at least of one such chamber, far ahead of the development of the corresponding space vehicles, so as to have these chambers available at the proper time (Bibl.25). The most ambitious project of this type is that of Mark II, projected by the AEDC; the scheduled dimensions are 75 m diameter and 90 m length (Bibl.14).

A review of these figures shows that, in itself, any generalization with respect to the future is quite impossible. It might well be, similar to the situation in aircraft construction, that large-scale tests will be limited by the dimensions of the spacecraft as well as by the practical experience and dependability obtained in the building of space vehicles. /23

The variety of simulator types, discussed here, does not exactly facilitate the selection at the time of projecting certain space programs.

Generally, equipment for simulation should be designed with consideration of the space vehicle in question and the nature of its mission; the characteristics of such equipment should be determined in collaboration with specialists among whom vacuum technologists, familiar with the problems of simulation, play an important role.

However, taking into consideration the generally rather short deadlines of the various phases of the programs and also financial aspects, satellite tests will mainly be performed in available simulators; for this reason, maximum use must be made of their performance, for determining the nature of additional tests to which the satellites must be submitted for defining their flight aptitude.

Thus, in practical application, the real function of simulators, combined with experimentation on a given satellite, will be a determination of the parameters to be simulated, including their values and their accuracy; this will lead to a certain testing and prediction pattern which will thus be doubly specific.

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